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British

(MALLEY)

415/115

#938,247 COMPLETE SPECIFICATION

1 SHEET

This drawing is a reproduction of the Original on a reduced scale.

Fig. 1.

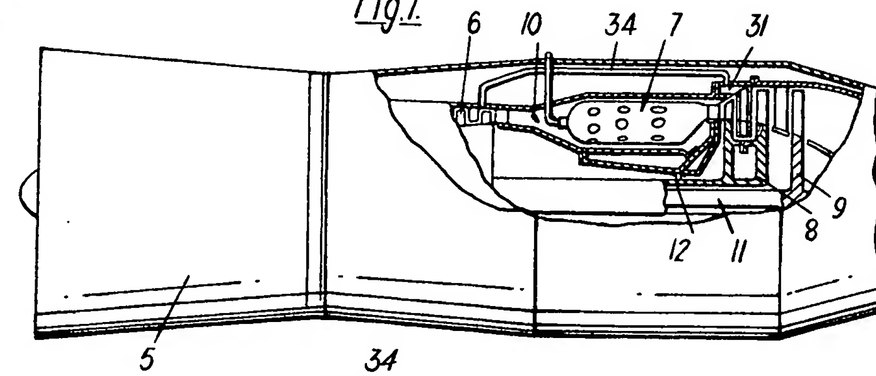
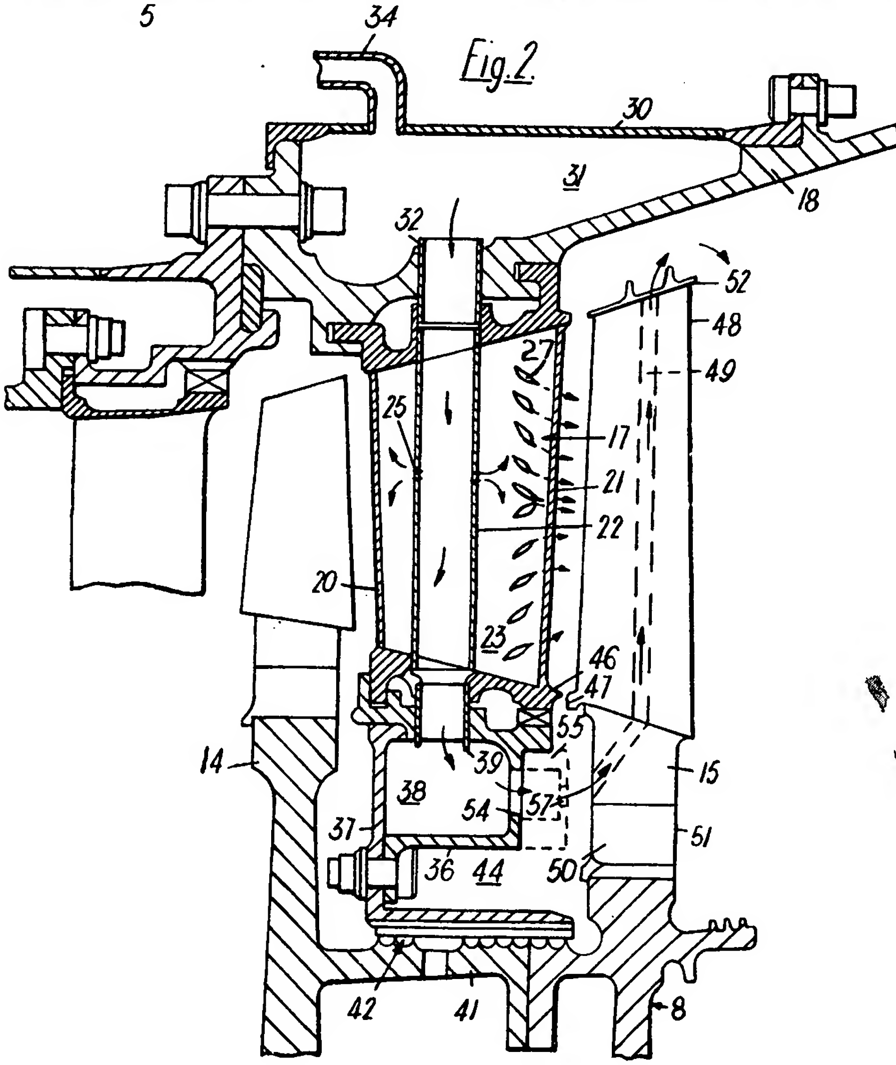


Fig. 2.



PATENT SPECIFICATION

DRAWINGS ATTACHED

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938,247

938,247

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Index at acceptance: —**Class 110(3), B2B2, G6.**

International Classification: —**F02c.**

COMPLETE SPECIFICATION

Gas Turbine Engine Having Cooled Turbine Blading

We, ROLLS-ROYCE LIMITED, a British company, of Nightingale Road, Derby, do hereby declare the invention, for which we pray that a patent may be granted to us, and the method by which it is to be performed, to be particularly described in and and by the following statement:—

This invention is concerned with cooling turbine rotor blading of a gas turbine engine.

If cooling air is tapped from a high pressure compressor of a gas turbine engine, and is fed internally via part of the engine disposed radially inwardly of the main gas duct of the engine, to one or more bladed turbine rotor stages of the engine for cooling the blades of said stage(s) large pressure losses may occur in the cooling air if it has to follow a tortuous path through the interior of the engine, and the cooling air may pick up a considerable amount of heat whilst flowing along said path, and so loose some of its ability to cool the blades of said stage(s). Where the cooling air cools first and second turbine rotor stages, and a turbine stator member is sealed against a turbine shaft section interconnecting said turbine rotor stages, the cooling air may by-pass the seal and reach the second turbine rotor stage through drillings in the turbine shaft. In one case it was found that such drillings, which were of the maximum size permitted in accordance with design considerations, gave rise to a pressure loss of some 40% of the cooling air supply pressure from the high pressure compressor. As a result, although the cooling of the first turbine rotor stage was satisfactory, the cooling of the second turbine rotor stage was inadequate.

According to one aspect of the present invention, there is provided a gas turbine engine having a main annular gas duct, at least one fixed hollow aerofoil member as herein defined spanning said duct, and means for supplying a cooling fluid radially inwardly across said duct via a passageway in said member to passages in blades of a bladed turbine rotor stage which is disposed in said duct downstream of said member, said cooling fluid in operation cooling said member and said blades, said member being disposed downstream of an additional bladed turbine rotor stage forming part of the same rotor as said first mentioned stage.

According to another aspect of the present invention, there is provided a gas turbine engine having a main annular gas duct, at least one fixed hollow aerofoil member as herein defined spanning said duct, means for supplying a cooling fluid to the radially outer end of said member, a passageway in said member for conveying said fluid radially inwardly across said duct, a bladed turbine rotor stage disposed in said duct downstream of said member, blades of said stage having cooling passages therein in fluid flow communication with the radially inner end of said passageway, cooling said member and said blades, sid memthrough said passageway and said passages and cooling said member and said blades, said member being disposed downstream of an additional bladed turbine rotor stage forming part of the same rotor as said first mentioned stage.

According to another aspect of the present invention, there is provided a gas turbine engine having a main annular gas duct, at least one fixed hollow aerofoil member as herein defined spanning said duct, means for supplying a cooling fluid to the radially outer end of said member, a passageway in said member for conveying said fluid radially inwardly across said duct, a blade turbine rotor stage disposed in said duct downstream of said member, the radially inner end of said member defining with said stage a portion of

[*Price* 4*s*. 6*d*.]

an annular chamber, said passageway communicating with said chamber, blades of said stage having passages therein also communicating with said chamber, cooling fluid in operation flowing successively through said passageway, said chamber and said passages, and cooling said member and said blades, said member being disposed downstream of an additional bladed turbine rotor stage forming part of the same rotor as said first mentioned stage.

By the term "fixed hollow aerofoil member" we mean a member which does not, in operation, rotate about the axis of the engine. The term may apply, for example, to a nozzle guide vane which may be pivotable.

The invention will be further explained, by way of example, with reference to the accompanying drawings in which:—

Fig. 1 is a partially cut-away side view of a gas turbine engine according to the invention, and

Fig. 2 is an enlarged longitudinal sectional view of part of Fig. 1.

The gas turbine engine shown in Fig. 1 comprises a low pressure compressor 5 (not shown in detail), a high pressure compressor 6, combustion equipment 7, a high pressure turbine 8, and a low pressure turbine 9 arranged in flow series in the main annular gas duct 10 of the engine. The compressor 5 and turbine 9 have a common shaft 11, and the compressor 6 and turbine 8 has a common shaft 12.

The turbine 8 comprises first and second bladed turbine rotor stages 14, 15 (see Fig. 2), and a plurality of angularly spaced fixed hollow aerofoil members in the form of nozzle guide vanes 17, only one of which is shown, the vanes 17 being disposed intermediate the stages 14, 15 axially of the engine. The vanes 17 are secured to an outer casing 18 of the duct 10, and span the duct 10.

Each vane 17 has leading and trailing edges 20, 21 respectively, and a tube 22 is mounted in each vane and extends longitudinally through the hollow interior of the vane, spaced from the internal surface thereof. Thus an internal chamber 23 is formed between the tube 22 and the internal surface of the vane, the chamber 23 wholly surrounding the circumference of the tube 22. Apertures such as 25 are formed in the tube 22, establishing communication between the interior of the tube and the chamber 23. Holes 27 of circular cross section are drilled obliquely through a concave wall of the vane adjacent the trailing edge 21 to establish communication between the chamber 23 and the duct 10.

The holes 27 may be formed as described in Specification 884,409.

A member 30 is secured around the casing 18 and forms an outer manifold 31 therewith. The tube 22 of vane 17 communicates with the interior of the manifold 31 via a respective connecting tube 32 mounted in the casing 18. A pipe 34 connects the manifold 31 to the compressor 6 as shown diagrammatically in Fig. 1.

Each vane 17 is connected at its radially inner end to an annular member 36 which forms with a further annular member 37 connected thereto an inner annular manifold 38. The tube 22 of each vane 17 communicates with the interior of the manifold 38 via a respective connecting tube 39 mounted in the member 36.

Between the member 37 and a shaft section 41 interconnecting the turbine rotor stages 14, 15 of the turbine 8, there is a labyrinth seal 42, and an annular chamber 44 is formed which is bounded by parts of the members 36, 37, the radially inner portion of the turbine rotor stage 15, and registering platforms 46, 47 provided on the vanes 17 and the turbine rotor stage 15. The latter includes a plurality of angularly spaced blades 48, in each of which there is a passage 49 extending from the forward face 50 of the blade root 51, to the radially outer tip 52 of the blade, where the passage 49 opens to the duct 10. It will be seen that the passages 49 communicate with the chamber 44.

The manifold 38 communicates with the chamber 44 via a plurality of angularly spaced apertures 54, which are formed in the member 36 at the same radial distance from the axis of the engine as the openings to the passages 49, so that the apertures 54 are opposite the openings to the passages 49.

A preferred optional feature is to mount a further annular member 55 on the member 36, which member 55 has a plurality of discharge orifices 57 therein whose axes are suitably inclined so that the orifices are adapted to direct air flowing therethrough into the passages 49 without substantial loss of pressure due to rotation of the turbine rotor 8.

In operation, compressed air from the compressor 6 flows via pipe 34, manifold 31, tubes 32, 22 and 39, manifold 38, apertures 54, discharge orifices 57, chamber 44 and passages 49 into the duct 10. A proportion of the air flowing through each tube 22 escapes through the apertures 25 into the respective chamber 23, and then flows through the holes 27 directly into the duct 10, the air fanning out over the trailing portion of the concave external surface of the respective vane 17. This proportion of air cools the respective vane 17 as it passes through the chamber 23 thereof, and effects film cooling of said trailing portion of the vane. The air flowing through the passages 49 cools the blades 48. The air in each chamber 23 thermally insulates the respective tube 22 so that relatively little heat passes inwardly of the vane 17 to heat the air which is flowing through the tube 22. Thus the ability of the air reaching the manifold 38 to cool the blades 48 of the tur-

bine rotor stage 15 is relatively unimpaired by crossing the duct 10.

The blades 48 can have internal passages other than those of the form which have been diagrammatically illustrated, to obtain efficient cooling of the blades 48 and desired operational temperature distributions chordwise and spanwise of the blades. Similarly the vanes 17 can be constructed in many different ways, and may for example have an integral construction and may be pivotable about their longitudinal axes.

The pipe 34 may contain a shut-off valve which may be operated automatically so that the blades are cooled only when they are operating at or above predetermined temperatures.

A heat exchanger or cooler may be provided in the pipe 34 in order to cool the air tapped from the compressor.

WHAT WE CLAIM IS:—

1. A gas turbine engine having a main annular gas duct, at least one fixed hollow aerofoil member as herein defined spanning said duct, and means for supplying a cooling fluid radially inwardly across said duct via a passageway in said member to passages in blades of a bladed turbine rotor stage which is disposed in said duct downstream of said member, said cooling fluid in operation cooling said member and said blades, said member being disposed downstream of an additional bladed turbine rotor stage forming part of the same rotor as said first mentioned stage.

2. A gas turbine engine having a main annular gas duct, at least one fixed hollow aerofoil member as herein defined spanning said duct, means for supplying a cooling fluid to the radially outer end of said member, a passageway in said member for conveying said fluid radially inwardly across said duct, a bladed turbine rotor stage, disposed in said duct downstream of said member, blades of said stage having cooling passages therein in fluid flow communication with the radially inner end of said passageway, cooling fluid in operation flowing successively through said passageway and said passages and cooling said member and said blades, said member being disposed downstream of an additional bladed turbine rotor stage forming part of the same rotor as said first mentioned stage.

3. A gas turbine engine having a main annular gas duct, at least one fixed hollow aerofoil member as herein defined spanning said duct, means for supplying a cooling fluid to the radially outer end of said member, a passageway in said member for conveying said fluid radially inwardly across said duct, a bladed turbine rotor stage disposed in said duct downstream of said member, the radially inner end of said member defining, with said stage a portion of an annular chamber, said passageway communicating with said chamber, blades of said stage having passages therein also communicating with said chamber, cooling fluid in operation flowing successively through said passageway, said chamber and said passages, and cooling said member and said blades, said member being disposed downstream of an additional bladed turbine rotor stage forming part of the same rotor as said first mentioned stage.

4. An engine as claimed in any preceding claim in which said first mentioned stage is a second turbine rotor stage.

5. An engine as claimed in claim 4 in which a high pressure turbine is disposed in said duct upstream of a separate low pressure turbine, said first mentioned stage being the second rotor stage of the high pressure turbine.

6. An engine as claimed in any preceding claim in which said member is adapted to discharge a proportion of the fluid flowing therethrough directly into said duct, which proportion serves for cooling said member.

7. An engine as claimed in claim 6 in which said member has an internal chamber into which said proportion of fluid flows after separating from the remainder of the fluid flowing through said member, said proportion of fluid escaping from the chamber into said duct.

8. An engine as claimed in claim 7 in which said internal chamber surrounds said passageway so that in operation fluid in said internal chamber thermally insulates the fluid flowing through said passageway.

9. An engine as claimed in any preceding claim in which said passages in said blades are open to said duct at the radially outer ends of said blades.

10. An engine as claimed in any preceding claim provided with a plurality of said members which are angularly spaced around said duct, the radially outer and inner ends of said passageways communicating respectively with a common outer annular manifold and a common inner annular manifold.

11. An engine as claimed in claim 10 in which said inner manifold has discharge orifice means adapted to direct fluid flowing therethrough into said pasages in said blades.

12. An engine as claimed in any preceding claim in which said means for supplying cooling fluid is arranged to receive air compressed by compressor means of the engine.

13. A gas turbine engine substantially as described with reference to and as shown in the accompanying drawings.

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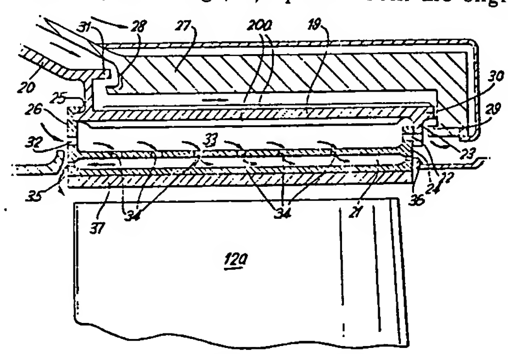
Blade tips seal for unshrouded gas turbine engine - has two annular sensing members of different initial rates of thermal expansion

ROLLS ROYCE 1971 LTD 07.12.74-GB-052998

(01.09.77) F01d-11/08

The unshrouded gas turbine engine has a blade tips seal consisting of a ring (21) spaced from the engine structure to form a sealing clearance therebetween. A first annular sealing member (19) is connected with the ring for movement therewith and a second member (27) cooperates with the first member for restrained movement therewith.



The first member has a higher initial rate of thermal expansion or contraction than the second member such that the first member will expand relatively more quickly with a temperature increase. The restraint will restrain the rate of contraction of the first member to that of the second member with a decrease in temperature. 3. 12. 75 (5pp1058).